2.0 MISSION DESIGN

2.1 Introduction

The goal of the Pre-Phase A mission design analysis effort was to identify the science and system design parameters that drive orbit selection and to develop a preliminary mission design. Previous work on the mission design had identified the launch energy envelope required to achieve the desired range of the spacecraft's relative drift rate to Earth. The proposed method for achieving the desired orbit was a direct insertion into heliocentric orbit [Reference 1]. One of the results of this work was the selection by the Science Definition Team of a two spacecraft formation. One spacecraft is placed into a heliocentric orbit that leads the Earth, while the second spacecraft follows behind the Earth in its orbit. Selection of the direct transfer mode identified a nominal launch energy, C3 = 1.0 km^2/sec^2. The scope of the current study is to identify the additional factors that impact the mission design. Factors considered for this study include single versus dual launch, launch window constraints, launch parameters and drift orbit selection.

2.2 Science Definition

The definition of the spacecraft orbit needed to fulfill the science objectives for the mission is derived from the recommendation of the Science Definition Team for one spacecraft to lead the Earth, with a second to follow the Earth with the following characteristics:

> "STEREO #1, leading Earth, will dwell near 20° between 200 and 400 days into the mission, and near 45° between 600 and 800 days. STEREO #2, lagging Earth, will dwell near 30° and 60°, respectively."

2.3 Solar Drift Orbit Mechanics

The heliocentric orbits selected by Solar TErrestrial RElations Observatory (STEREO) mission are well represented by the classical Keplerian orbital elements of Semi-major axis α, Eccentricity e, Inclination i, Right Ascension of the Ascending Node Ω, Argument of Perihelion ω, and True Anomaly ν, in the heliocentric reference frame. The science definition is primarily concerned with the semi-major axis because it directly determines the mean drift rate relative to Earth, and is the primary factor in determining the dwell time history.

A convenient mapping of the heliocentric orbit into the departure conditions from the Earth is provided by the Zero Sphere of Influence Patched Conic Model. In this model the departure condition is simply defined by the V_∞ vector as illustrated in Figure 2-1. V_∞ is the vector difference between the velocity of the spacecraft's heliocentric orbit and the velocity of the Earth. The magnitude of V_∞ is referred to as the hyperbolic excess speed. The equation $C3 \equiv |V_\infty|^2$ relates the constant C3, to the hyperbolic excess speed. The escape angle α is defined as the angle between the Earth's velocity direction and V_∞ as shown in the figure. Although the figure shows the vectors drawn in the Earth's orbit plane; selected pairs of V_∞ and α actually represent a locus of solutions which describe a cone of half-angle a around the Earth's velocity vector.

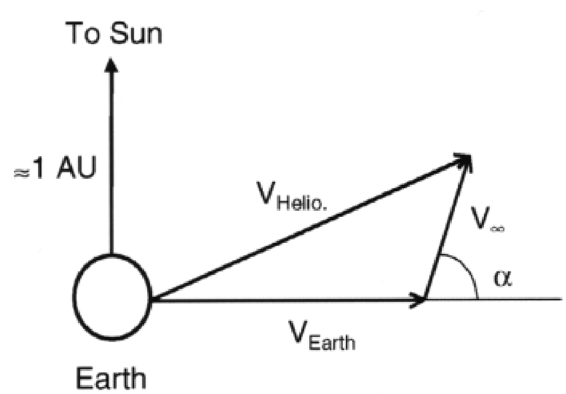


Figure 2-1 Earth Escape Parameters

Neglecting the small effect of the Earth's orbital eccentricity (0.017), it can be shown that the spacecraft's heliocentric semi-major axis, and therefore mean drift rate, is only a function of the spacecraft's heliocentric velocity. From this we are able to parameterize all heliocentric drift orbits by C3 and α. For the energies of interest to STEREO, (e.g., C3 $\approx$ 1.0 km^2/sec^2) the trajectory design space can be described by a contour plot showing the mean drift rate relative to Earth, η as a function of C3 and α as shown in Figure 2-2. A positive drift rate defines a leading orbit, negative a lagging orbit. In the scenario where the spacecraft is inserted directly into a heliocentric orbit by the launch vehicle, two important features can be identified from this mapping. First, for a selected mean drift rate, increasing the C3 value lessens the sensitivity of the drift rate to C3 variations. This is shown by the fact that contours of constant drift rate become nearly vertical as C3 increases. C3 variations are the major error source from the launch vehicle.

A second, somewhat contrary feature is that for a given mean drift rate, selecting the C3 value near the minimum C3 lessens the sensitivity of the drift rate to launch time. This is shown by the fact that contours of constant drift rate become nearly horizontal near the minimum C3. This assertion is made because we can assume that the rate of change of α is proportional to the Earth's rotation rate for most launch scenarios. The exception is a launch vehicle that can fly a variable azimuth trajectory as a function of launch time.

The mapping of C3 and α into the mean drift rate shown in the figure is idealized because we've used the Zero Sphere of Influence Patched Conic Model. This approach was selected to permit a closed form solution. The mapping also exists in more complex models that more fully

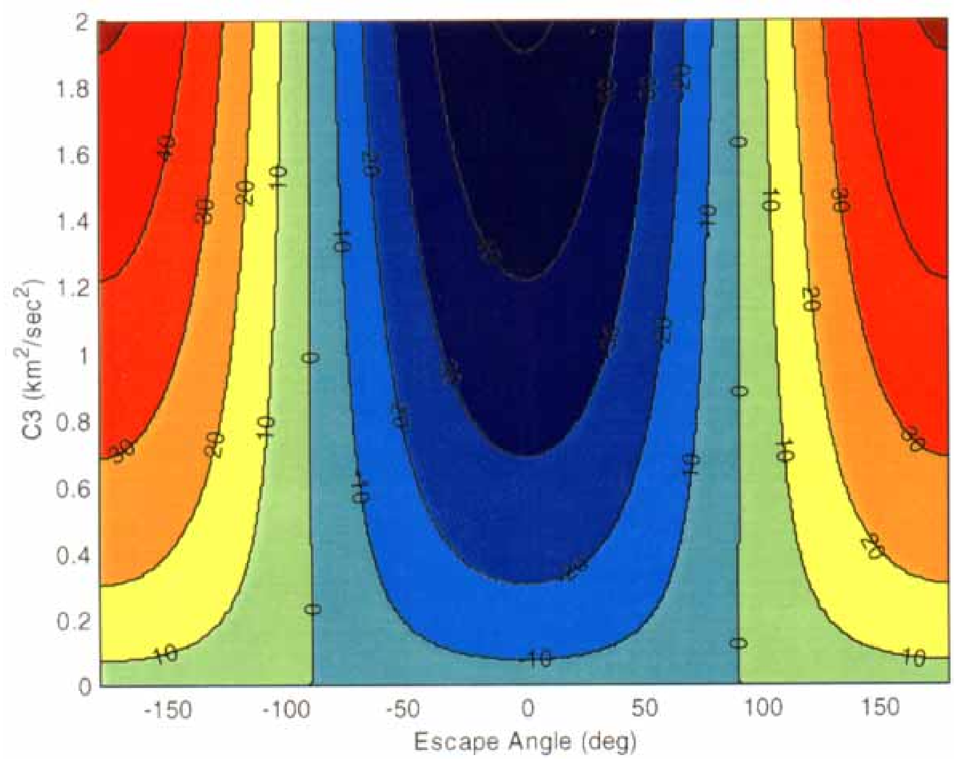


Figure 2-2 STEREO Mission Design Space

account for other perturbations. In these models the mapping retains it basic topology, but becomes increasingly distorted for low energy trajectories with escape angles approaching ±90°. The design point for most STEREO trajectories considered for this study, as well as the nominal design to be presented are far from this region.

All the trajectories for this study were computed in a complete solar system model that includes point mass gravitational effects of the Sun, Earth, Venus, Mars, Jupiter, and Saturn for the heliocentric phase. The heliocentric phase is defined as spacecraft to Earth distances > 900,000 km. For smaller distances, the gravitational effects of the Earth, Sun and Moon were used to model the spacecraft motion. The Earth was modeled with a 4×4 gravity field. The Sun and Moon are modeled as point masses. Perturbations due to solar radiation pressure were used in modeling both mission phases.

2.4 Families of Solar Drift Orbits

The primary interaction of the science definition and the orbital mechanics is the mean drift rate, which directly determines the semi-major axis. Figure 2-2 shows that there are actually families of drift orbits distinguishable by sets of the ordered pair (C3, α). One important family of trajectories are those that represent the minimum C3 (α = 0°, 180°) for a given drift rate. Ideally, these solutions correspond to maximum payload mass. Another trajectory family are those orbits with an $i = 0$. These planar solutions have their V_∞ in the ecliptic plane. For this report, the nominal mission trajectories were not restricted to any particular family of solutions, but utilized a full three-dimensional parameterization to best meet the science definition and optimize the system design.

Figure 2-3 and 2-4 show an ecliptic plane projection of a leading and lagging trajectory with a mean drift rate of +30 and −30°/year, respectively. A number of important parameters that impact the system design are derived from each trajectory. Figure 2-5 shows the spacecraft-Sun distance as a function of time (power, thermal). The spacecraft-Earth distance is shown in Figure 2-6 (telecommunications). Permutations of Sun, Probe, Earth angles are given in Figures 2-7 through 2-9 (telecommunications).

A principal concern of this study is the Sun-Probe-Earth (SPE) angle, which defines the antenna gimbal limits for the High Gain Antenna system for the spacecraft's nominal Sun pointing attitude. For the purposes of this study, the gimbal angle is equal to the SPE angle, where 0° corresponds to conjunction of the Sun and Earth as seen from the spacecraft. Figure 2-9 shows that for the leading trajectory the SPE angle is greater than 90° for about the first 200 days of the mission. The maximum value is approximately 165° after approximately 75 days. This is a general characteristic of leading trajectories. Both the maximum value and duration above 90° have been identified as major design drivers for the telecommunications system.

2.5 Transfer Trajectory and Launch Mode

Three types of transfer trajectories have been identified for possible use by STEREO to achieve the desired solar drift orbit. The choice of transfer trajectory is coupled to the possibility of launching the spacecraft independently (single launch) or together (dual launch). The three types of transfer trajectories are lunar flyby, libration point phasing, or direct insertion. The key ingredient to the lunar flyby trajectory is the use of a single or multiple flybys of the Moon in order to achieve the desired V_∞. In general, this type of trajectory requires the lowest C3, and therefore yields the highest payload mass of the three options. Spacecraft propulsion is required to accurately achieve the desired lunar flyby conditions. One or two months in a phasing

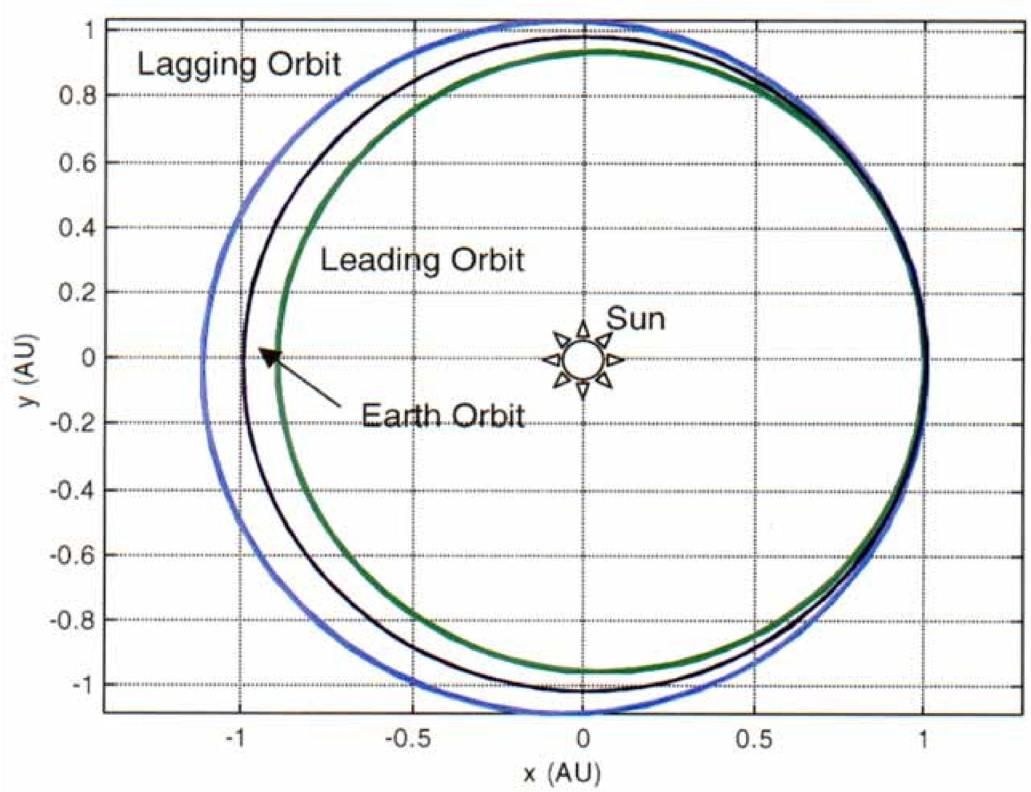


Figure 2-3 Sample Orbit, Heliocentric View

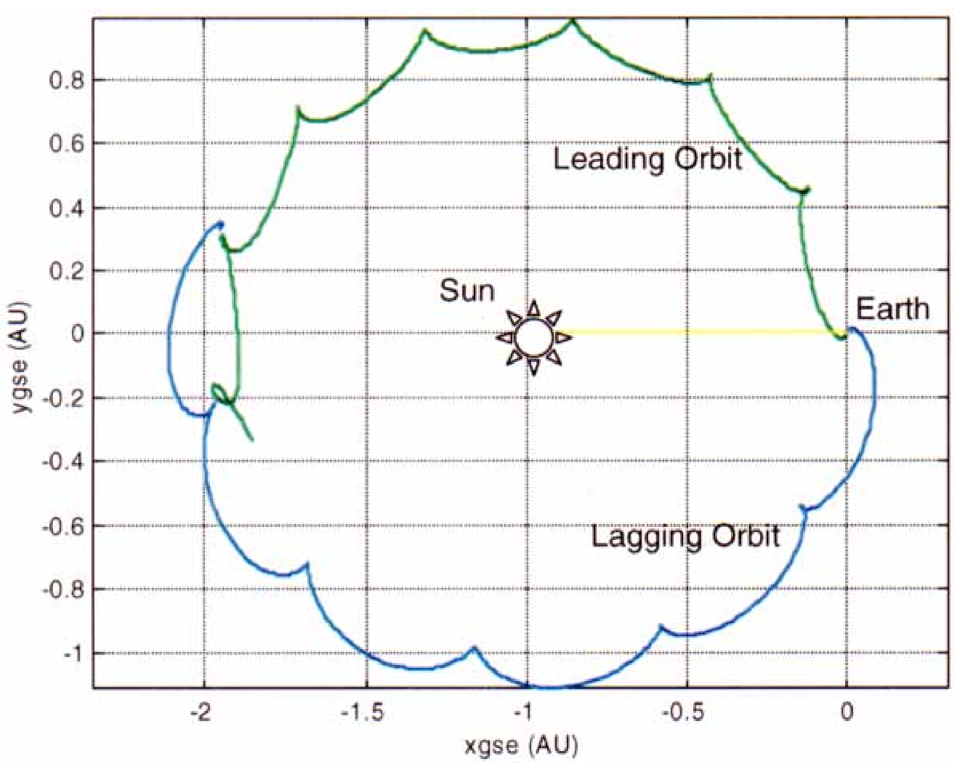


Figure 2-4 Sample Orbit, Geocentric Solar Ecliptic View

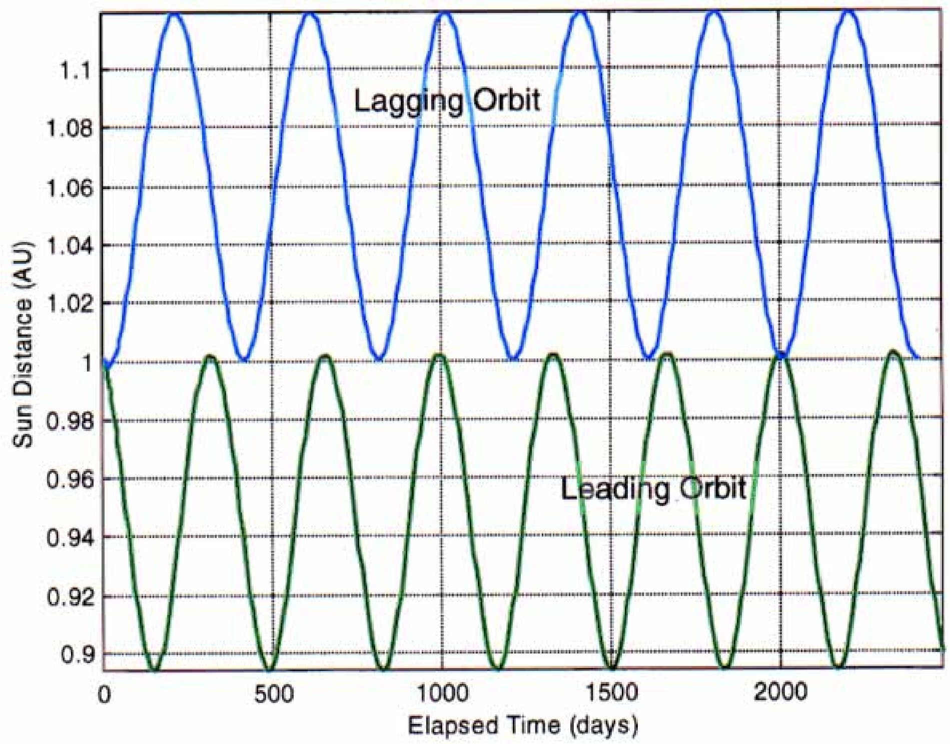


Figure 2-5 Sample Orbit, Sun Distance

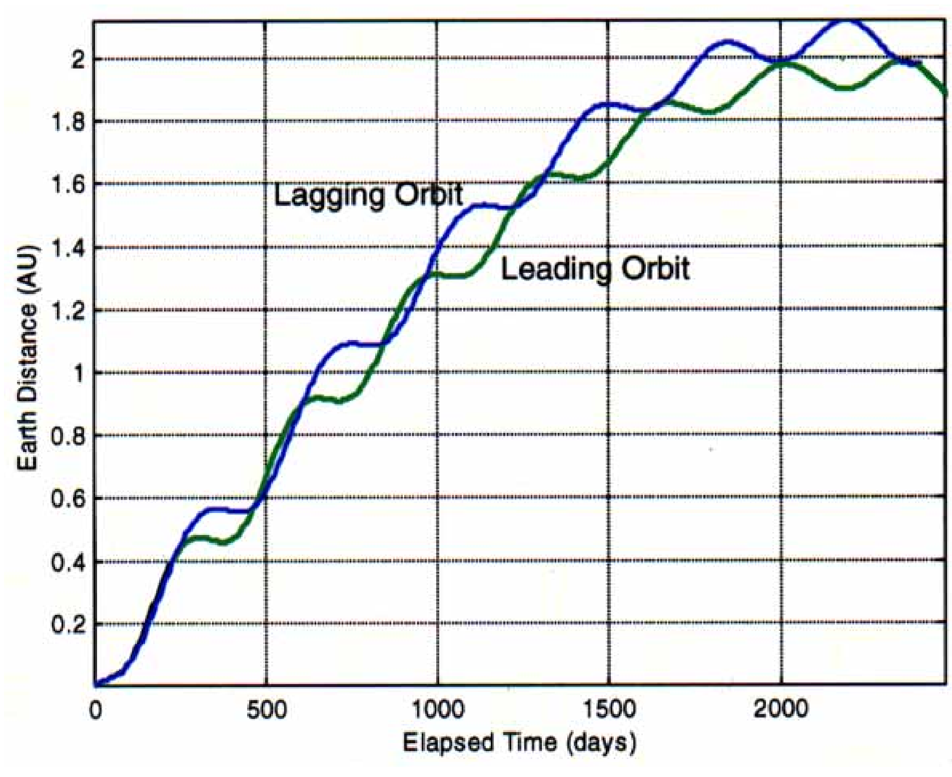


Figure 2-6 Sample Orbit, Earth Distance

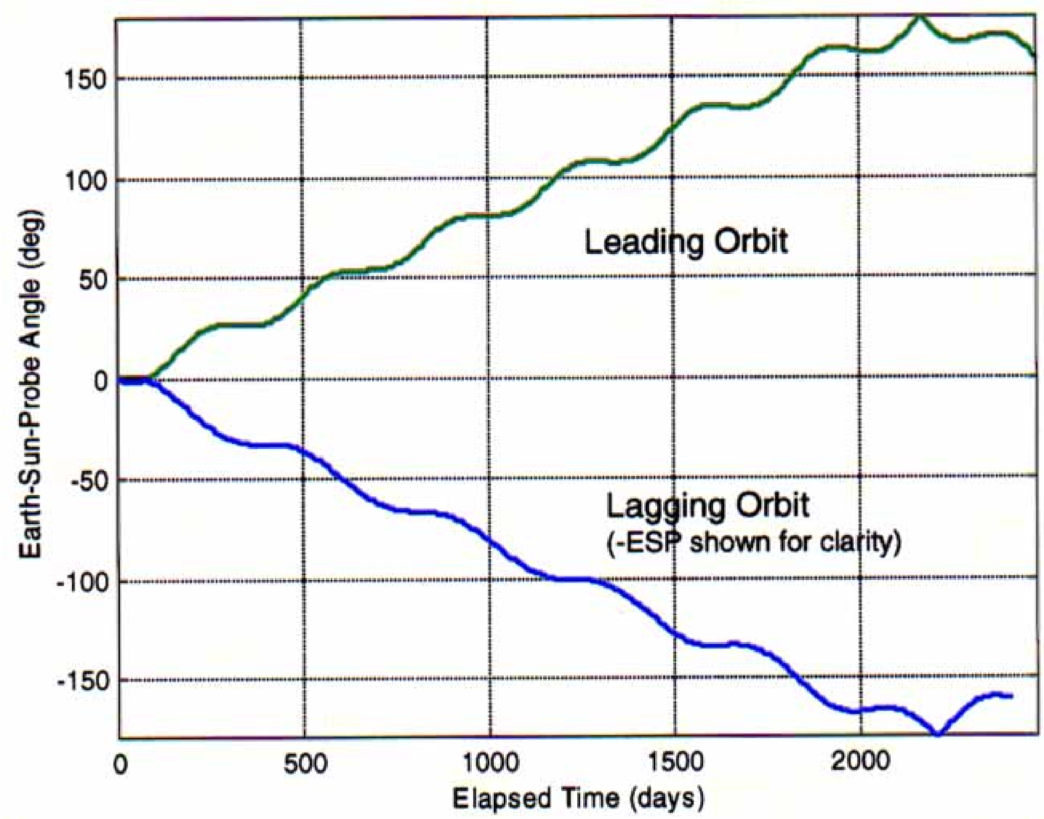


Figure 2-7 Sample Orbit, ESP Angle

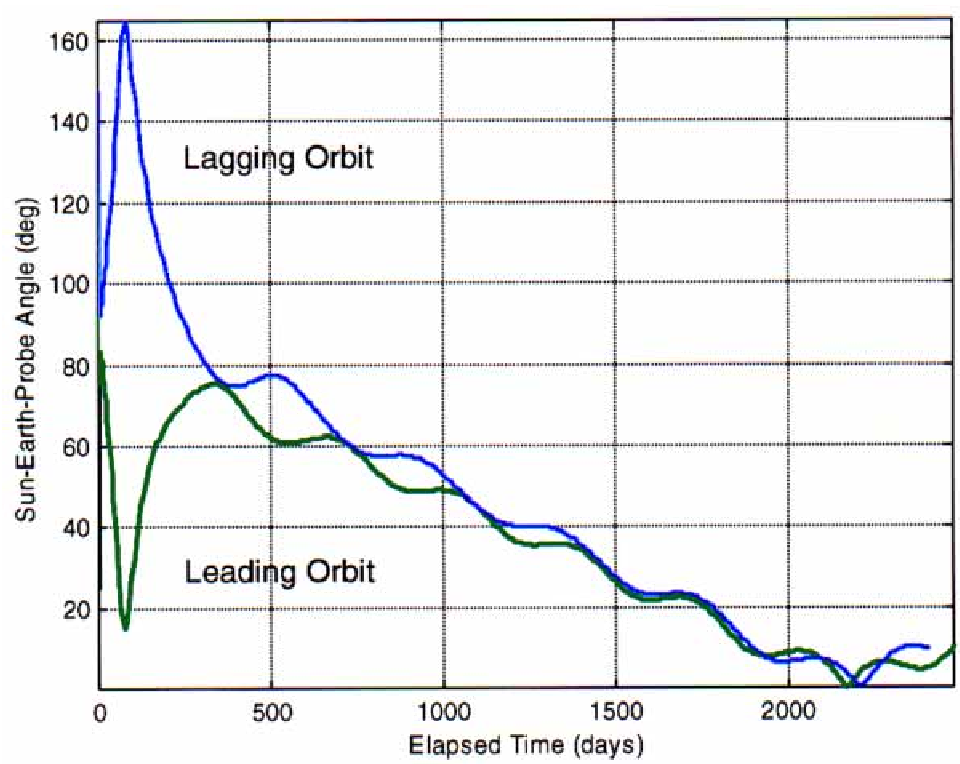


Figure 2-8 Sample Orbit, SEP Angle

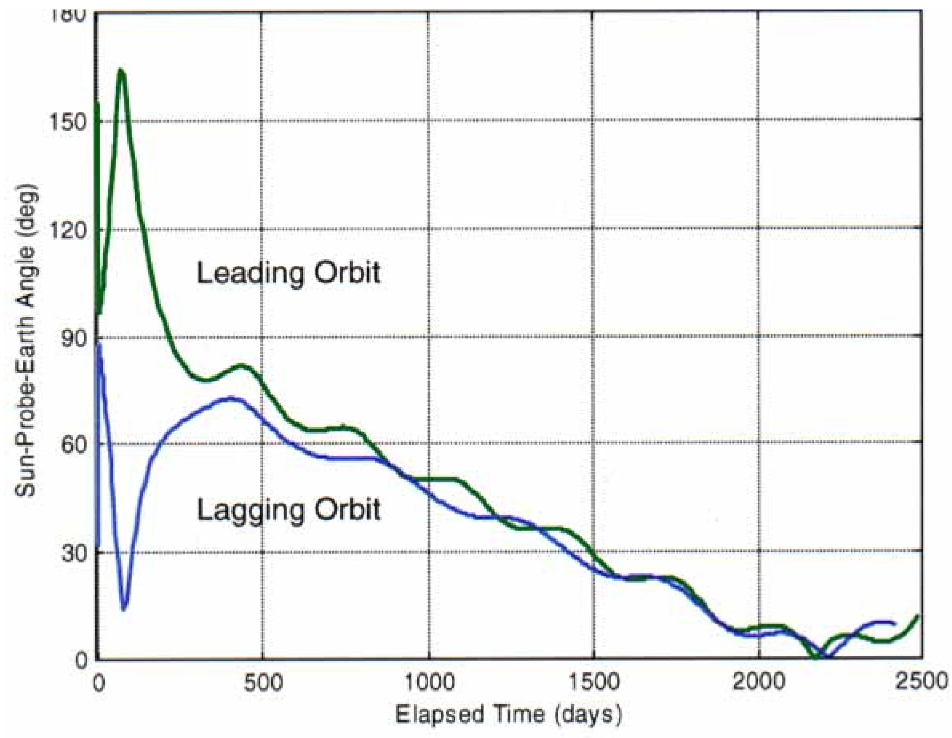


Figure 2-9 Sample orbit, SPE Angle

orbit prior to the first lunar flyby is desirable in order to decrease spacecraft fuel requirements and provide a reasonable number of launch opportunities per month.

The libration point transfers feature an excursion to either one or both of the interior or exterior libration points. The libration points are designated L_1 and L_2 for the interior and exterior points, respectively. They are located on the Sun-Earth line, approximately 1.5 million kilometers from Earth. Libration point transfers have an intermediate C3 requirement when compared to lunar flyby or direct insertion trajectories. The excursion to the libration point will typically consume more than seven months prior to Earth escape. Libration point transfers are well suited for a dual launch scenario where flexibility in selecting the final heliocentric orbit of each spacecraft independently is highly desirable. Spacecraft propulsion is required to control the orbit prior to escape from the Earth. Launch opportunities are generally available most days of the year, with a few days excluded each month due to undesirable lunar perturbations. Desirable lunar perturbations (i.e., lunar flybys) can be combined with libration point phasing to offer a high mass, high flexibility trajectory. The International Sun Earth Explorer (ISEE-3)/ International Cometary Explorer (ICE), Wind, Solar and Heliospheric Observatory (SOHO), and ACE missions have aptly demonstrated the utility of lunar flyby and libration points orbits. The Microwave Anisotrophy Probe (MAP) and Genesis missions have also selected these transfer types as their baseline mission design.

Direct insertion into heliocentric orbit offers the simplest approach to the STEREO mission design. This approach is well suited for the single launch mode. When the launch vehicle has sufficient lift mass, a dual launch can be accomplished. For a typical expendable launch vehicle (ELV) such as the Taurus, Athena-II, or Delta-II there is a significant impact to the flexibility of heliocentric orbit selection if a dual launch were used. A dual launch on the Space Shuttle offers more flexibility in orbit selection because of the extended mission duration (days vs. 1.5 hours). The preferred scenario for an ELV is the single launch mode. Up to one complete revolution in a low Earth parking orbit is required to fully exploit the trajectory design space shown in Figure 2-2. Although, if required by launch vehicle mass limits a direct ascent by the launch vehicle is a feasible means for achieving heliocentric orbit. No spacecraft propulsion is required for direct insertion transfers. This allows for a less complex spacecraft design and may negate the mass advantage of the lunar flyby and libration point trajectories. By definition, the heliocentric orbit is established once the spacecraft leaves the low Earth parking orbit less than two hours after launch. Launch opportunities are generally available most days of the year. A direct insertion approach was selected as the transfer trajectory for this study.

2.6 Launch Window

Two types of launch windows are defined for STEREO. The first, called the Launch Opportunity Window is defined as the days on which the spacecraft can be launched. The second, is the Daily Launch Window which is defined as the time of day you can launch. In order to discuss either type, we must first make the connection between the Earth escape parameters, represented by V_∞ and the launch vehicle and parking orbit parameters. A convenient parameterization of V_∞ is shown in Figure 2-10. The angles β and Δ are measured from the Earth's velocity direction. The angle β is a right-handed rotation around the Ecliptic pole and describes the offset in right ascension of V_∞ from the reference direction. For lagging trajectories the reference direction is along the Earth's velocity vector, for leading trajectories the reference direction is opposite the Earth's velocity direction. The angle Δ is the declination of V_∞ to the Ecliptic plane. The escape angle α, or it's supplement, is the hypotenuse of a spherical right triangle with sides β and Δ. The minimum energy trajectory for a selected mean drift rate corresponds to $\beta = \Delta = 0$. This particular parameterization allows for a straightforward mapping of V_∞ into the heliocentric orbital elements.

A second parameterization of V_∞ is the right ascension and declination of the vector relative the Earth's equator. These parameters, along

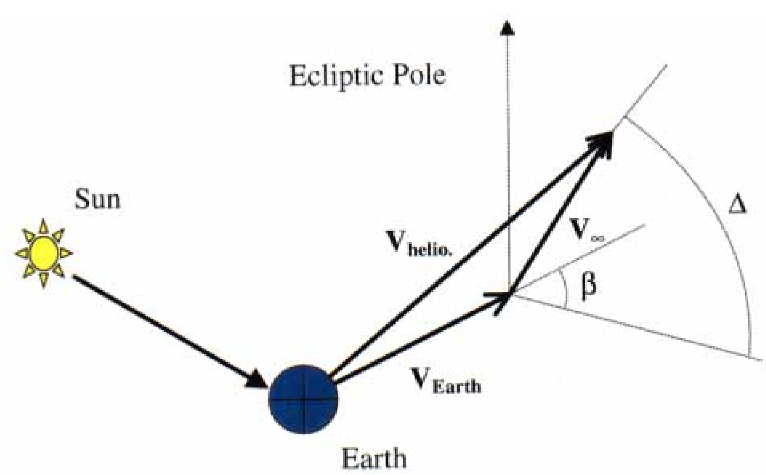


Figure 2-10 Design Parameterization

with the vector magnitude provide the geometric constraints needed to determine the launch time and parking orbit coast times for any given launch date. In general there are two launch opportunities per day, each with an associated parking orbit coast time. It is anticipated that only one of the two daily opportunities will be used. Since both opportunities result in the same heliocentric orbit, other factors such as station visibility, eclipse conditions, and launch time will be used as selection criteria between the two.

A significant constraint to any Earth escape mission is that the geocentric declination δ, of V_∞ can not exceed the parking orbit inclination for a planar injection. The launch site location and the launch azimuth determine the range of available parking orbit inclinations. Two factors impact the selection of launch azimuth. First, range safety limits the available azimuth range. Second, the maximum payload mass is achieved using a launch azimuth of 90° to take maximum advantage of the Earth's rotation.

2.6.1 Launch Opportunity Window

The minimization of the SPE angle becomes a factor in determining the Launch Opportunity Window (LOW) because of constraints imposed on the selection of Δ. The selection of a nonzero value of Δ results in an inclined heliocentric orbit. The spacecraft motion out of the ecliptic plane in an inclined orbit reduces the maximum SPE angle as shown in Figure 2-11. The value of the maximum SPE angle is related to all three of the design parameters (V_∞, β, Δ), but is most strongly dependent on Δ. In general, increasing Δ reduces the maximum value of the SPE angle for the leading spacecraft. Figure 2-12 shows the relationship between the time of year and maximum Δ. The figure shows the right ascension and declination of the Earth's velocity vector, which is our reference direction over the course of a year. The limiting case occurs at the equinoxes. Recall that the Earth's velocity is 90° out of phase with the Sun, so that the maximum declination occurs at the equinox rather than the solstice, while the zero crossings occur at the solstices. Also, recall that the maximum geocentric declination of V_∞ is a function of the parking orbit inclination, which in turn is determined by the launch azimuth and launch site latitude.

Two cases of interest to STEREO are an ELV or Shuttle launch from the Eastern Range. Assuming the ELV flies a maximum payload mass trajectory by launching due East (Launch Azimuth, AZ = 90°) the corresponding parking orbit inclination is 28.5°. Therefore, the maximum geocentric declination for V_∞ is also 28.5°. If the launch takes place near the equinox, the maximum Δ of 52°

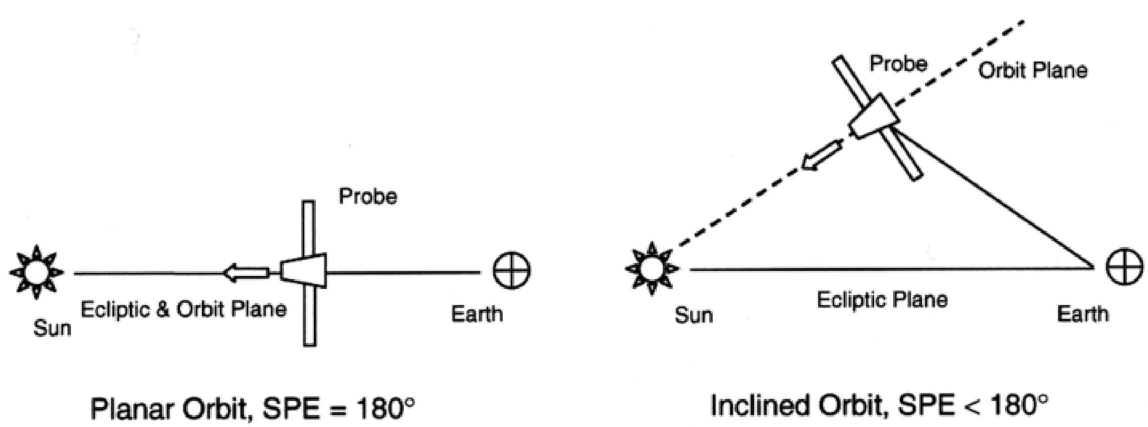


Figure 2-11 SPE Angle Minimization for Inclined Orbits

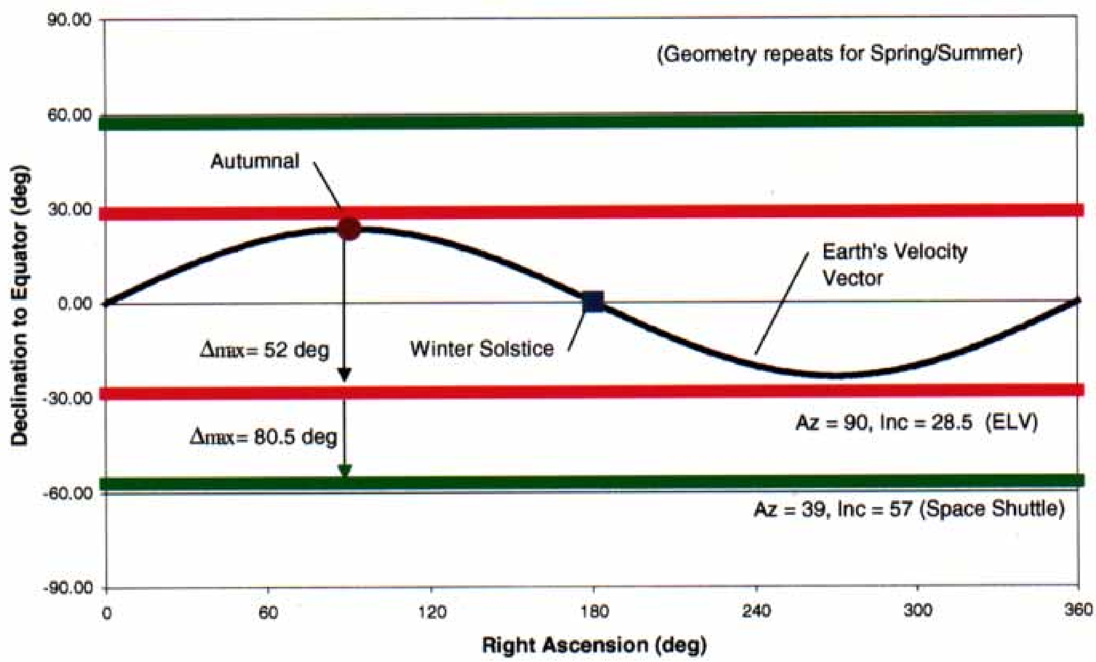


Figure 2-12 Launch Opportunity Window Trades

may be achieved. For a launch near the solstice, the maximum Δ is smaller. Since the higher Δ values are desired to reduce the maximum SPE angle for the leading spacecraft, launch near the equinox is preferred. If we consider a launch using the Space Shuttle, and assume no payload mass penalty for a high inclination parking orbit, the maximum value of Δ increases to 80.5°. Analysis performed for this study shows a 10° to 20° reduction in the maximum SPE angle for a Space Shuttle launch using a parking orbit inclination of 57°. A more detailed analysis of the impact of launch date on the SPE angle will be performed during Phase A.

2.6.2 Daily Launch Window

The primary drivers for determining the daily launch window are the early mission geometry with respect to the Sun and Earth, parking orbit coast time, and the sensitivity to launch time. As mentioned previously, there are typically two launch opportunities per day that achieves the same V_∞, and therefore the same heliocentric trajectory. The two launch opportunities have different launch times and parking orbit coast times. In some cases, the desired escape direction may require a full revolution in the spacecraft parking orbit. Long parking orbit coast times impact both the launch vehicle and spacecraft. The launch vehicle may require additional battery lifetime or expendables to coast for an entire orbit. The parking orbit coast time is a significant factor in the battery sizing for STEREO. Spacecraft power and thermal requirements may also impose additional launch vehicle requirements for attitude control during the coast phase. The parking orbit coast time also determines the post-heliocentric orbit insertion spacecraft-to-station geometry. The combination of launch and parking orbit coast times determines the spacecraft-Sun geometry, which determines the nature of spacecraft eclipse events.

The length of the daily launch window may be determined be examining the sensitivity of the heliocentric trajectory to the launch time.

Assuming the launch vehicle performs the same ascent trajectory over the daily launch window the primary effect on the trajectory is dispersion of the mean drift rate of the heliocentric orbit. A numerical estimate of this sensitivity can be obtained using Figure 2-2. For trajectories of interest to STEREO, ($C3 \cong 1$ km^2/sec^2, $\alpha \cong 50°$) the sensitivity in drift rate is approximately 1° per year for an 8-minute launch window, assuming the change in α is proportional to the Earth's rotation rate. This result has been verified for the nominal trajectory design presented below. The final determination of the daily launch window will be based on the launch vehicle error analysis performed for the STEREO configuration and the desired tolerance on the mean drift rate.

2.7 Nominal Trajectory Design

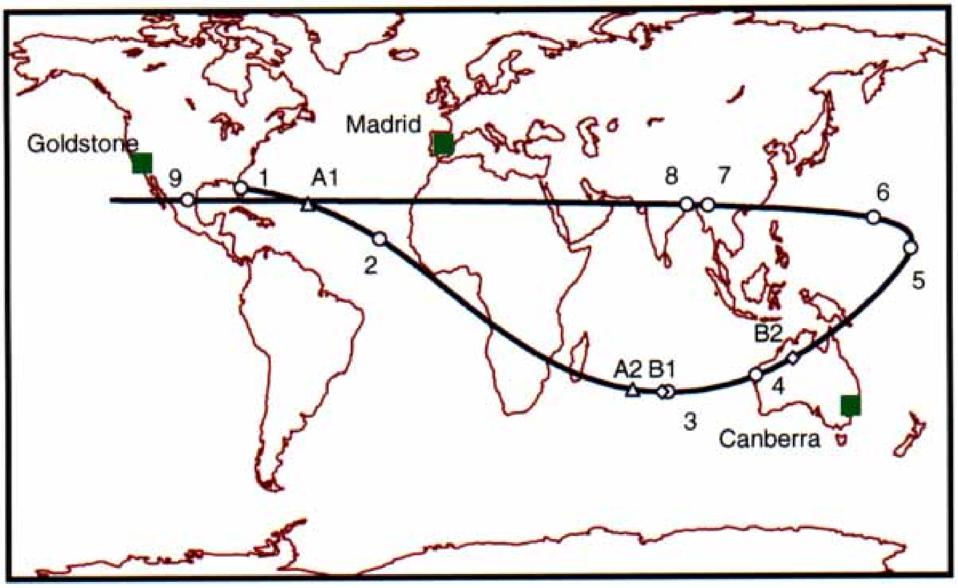
The science definition and mission design drivers that are cited above have all been considered in developing a nominal trajectory design for the STEREO mission. The mission design is based on two single spacecraft launches, sixty days apart aboard an Athena-II ELV from the Eastern Range. The spacecraft will be placed directly into heliocentric orbit from a low Earth parking orbit. A preliminary parking orbit definition was provided by Lockheed-Martin for the Athena-II based on a launch azimuth of 93°. The targeted mean drift rates are 20° and 28° per year for the leading and lagging spacecraft, respectively. No spacecraft propulsion is required to achieve or maintain the mission orbit. A nominal C3 of 1.0 km^2/sec^2 was selected for both spacecraft. This launch energy is greater than the minimum required for the mean drift rates in order to reduce the sensitivity to launch vehicle energy dispersions. The trajectory design parameters are given in Table 2-1. The leading spacecraft, which has a lower mean drift rate, is launched first in order to minimize the impact on the orbital formation of any delays in the launching of the second spacecraft. The launch date for the leading spacecraft is also closer to the equinox. For a given values of η and C3, α is fixed. For the leading trajectory, the values of β and Δ were selected to minimize the SPE angle during early mission and place the location of the maximum SPE angle as close to Earth as possible. Their values are subject to the constraints that α, β, and Δ form a spherical right triangle and geocentric declination of V_∞, $\delta \leq$ parking orbit inclination. The design parameters for the lagging trajectory were selected so that the launch phases for each spacecraft are identical. Figures 2-13 through 2-21 present the nominal mission design.

2.8 References

Reference 1: *The Sun and Heliosphere in Three Dimensions: Report of the NASA Science Definition Team for the STEREO Mission*, December 1997.

Table 2-1 - STEREO Mission Design Parameters

Parameter	Leading (STEREO-1)	Lagging (STEREO-2)
Launch Date	October 1, 2002	December 1, 2002
η (deg/yr)	20	−28
C3 (km2/sec2)	1.0	1.0
α (deg)	60	45
β (deg)	−41	30
Δ (deg)	49	35
δ (deg)	28	28



No.	Time	Event	No.	Time	Lighting Event
1	L + 0 min	Launch		*STEREO-1*	
2	L + 16 min	Park Orbit Insert.	A1	L + 10 min	Enter Eclipse
3	L + 49 min	Helio. Orbit Insert.	A2	L + 45 min	Exit Eclipse
4	L + 56 min	Canberra AOS			
5	L + 2.0 hr	Goldstone AOS		*STEREO-2*	
6	L + 4.3 hr	Goldstone LOS	B1	L+ 48 min	Enter Eclipse
7	L + 9.2 hr	Canberra LOS	B2	L + 60 min	Exit Eclipse
8	L + 9.9 hr	Madrid AOS			
9	L + 22.9 hr	Madrid LOS			

Figure 2-13 Early Mission Groundtrack

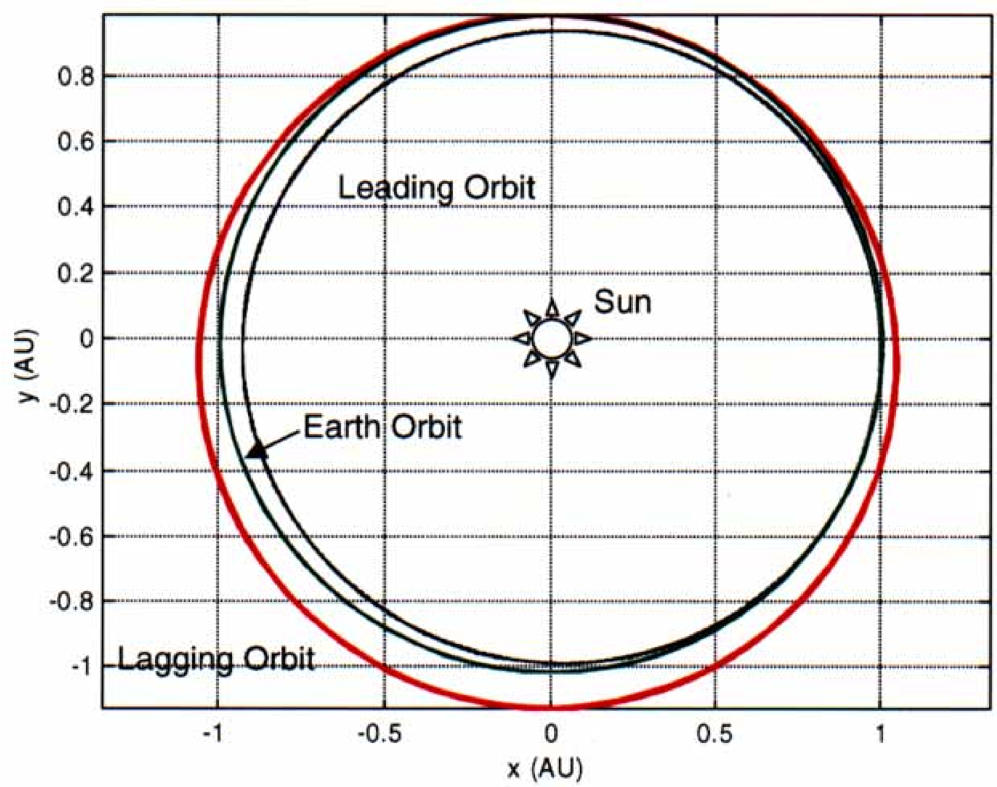


Figure 2-14 Nominal Heliocentric View

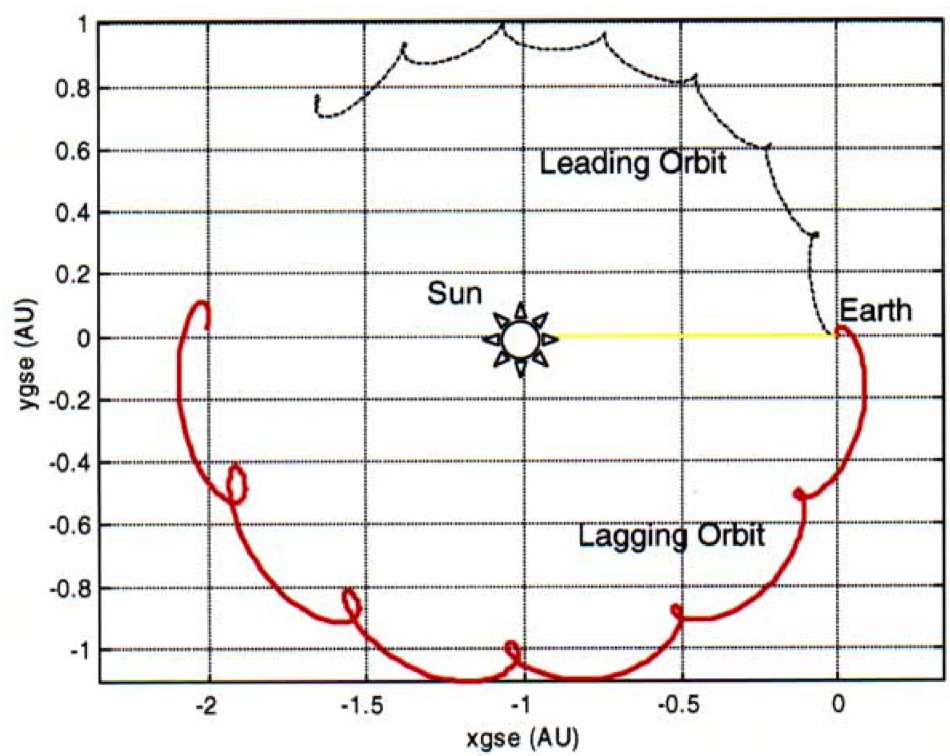


Figure 2-15 Nominal Geocentric Solar Ecliptic

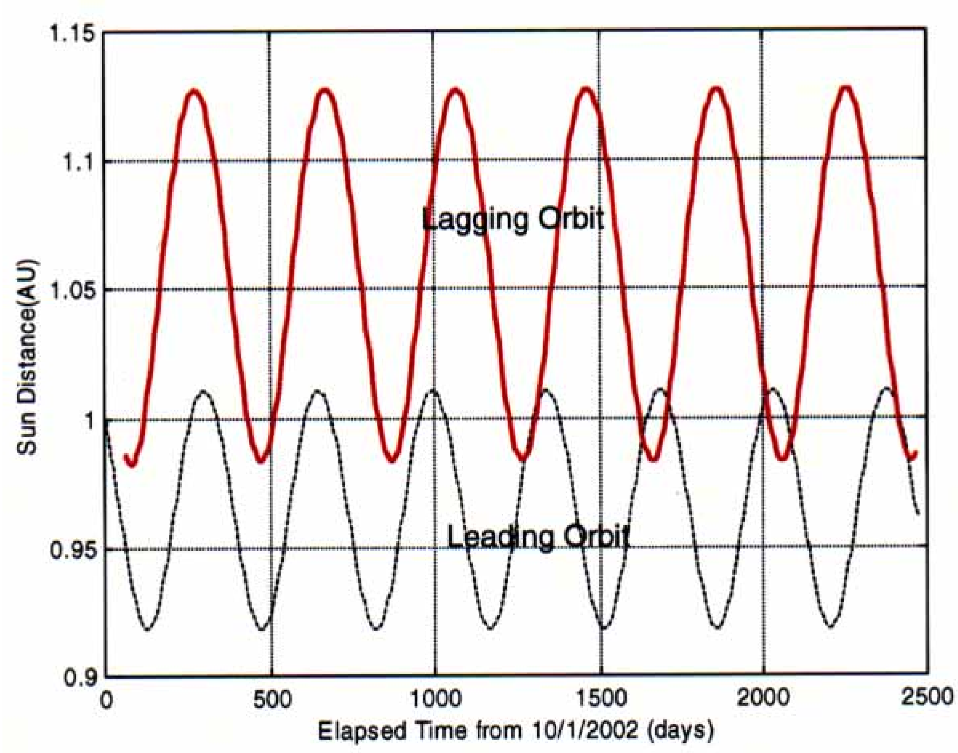


Figure 2-16 Sun Range

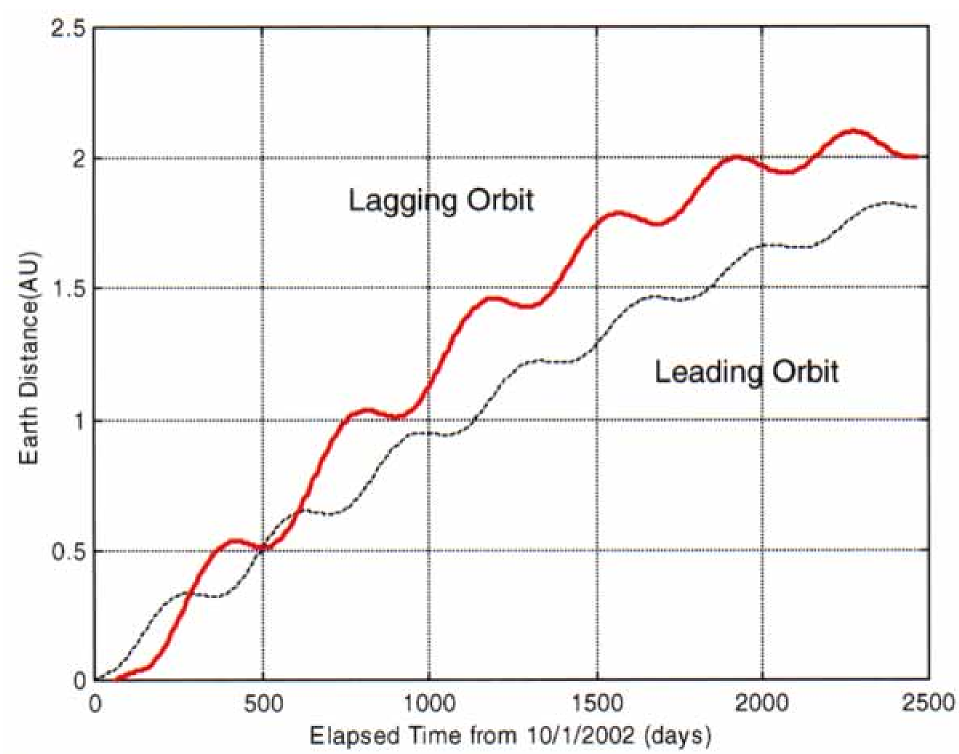


Figure 2-17 Earth Range

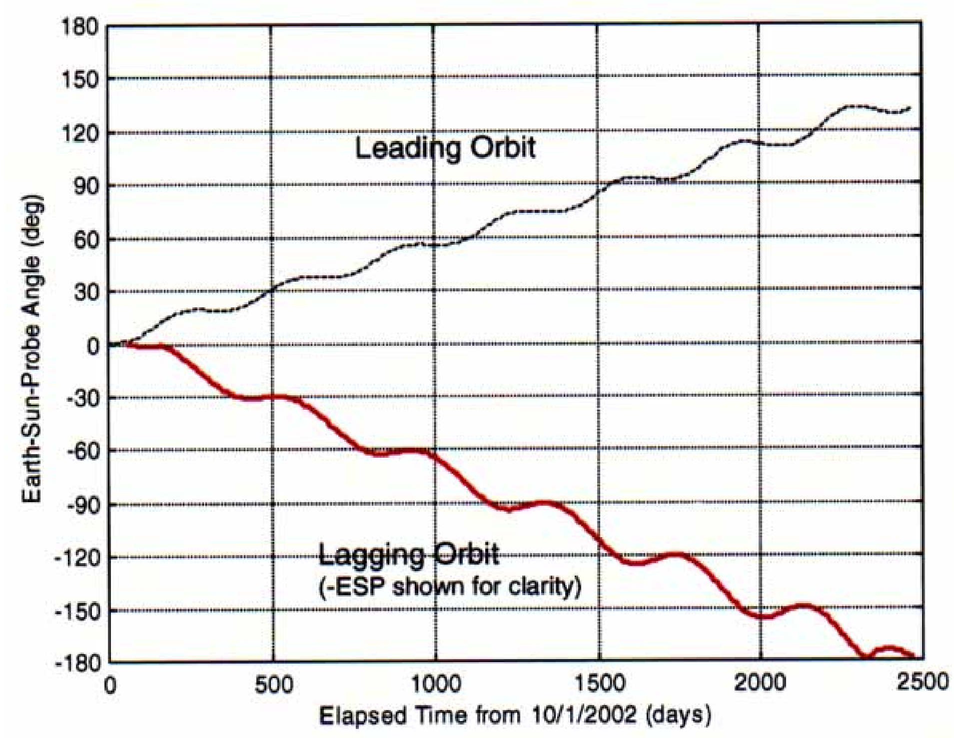


Figure 2-18 Earth-Sun-Probe Angle

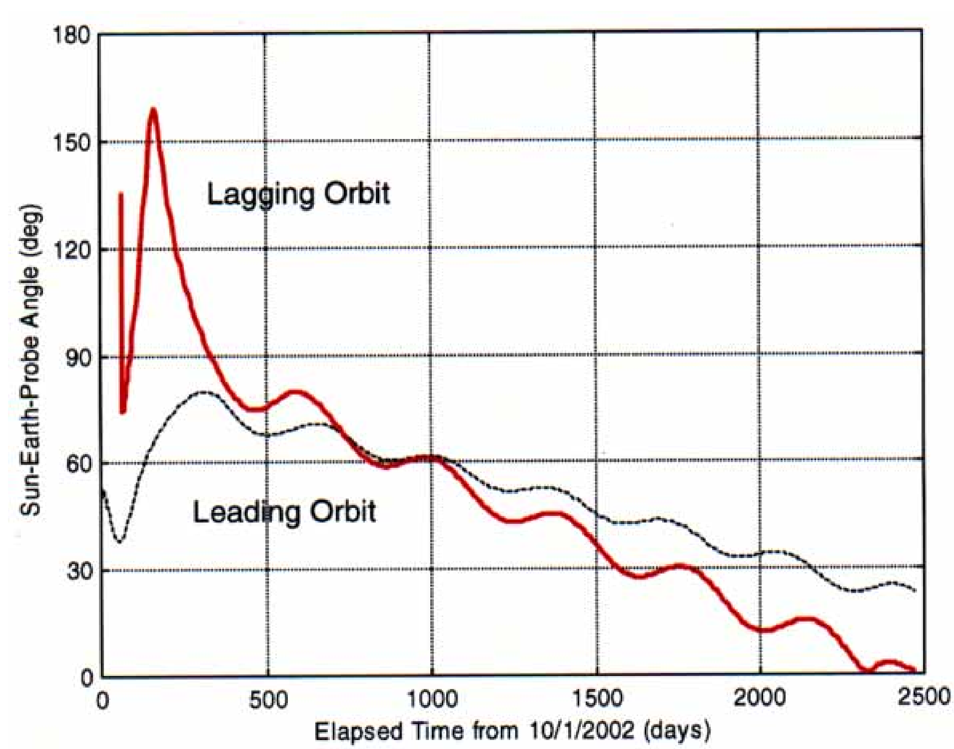


Figure 2-19 Sun-Earth-Probe Angle

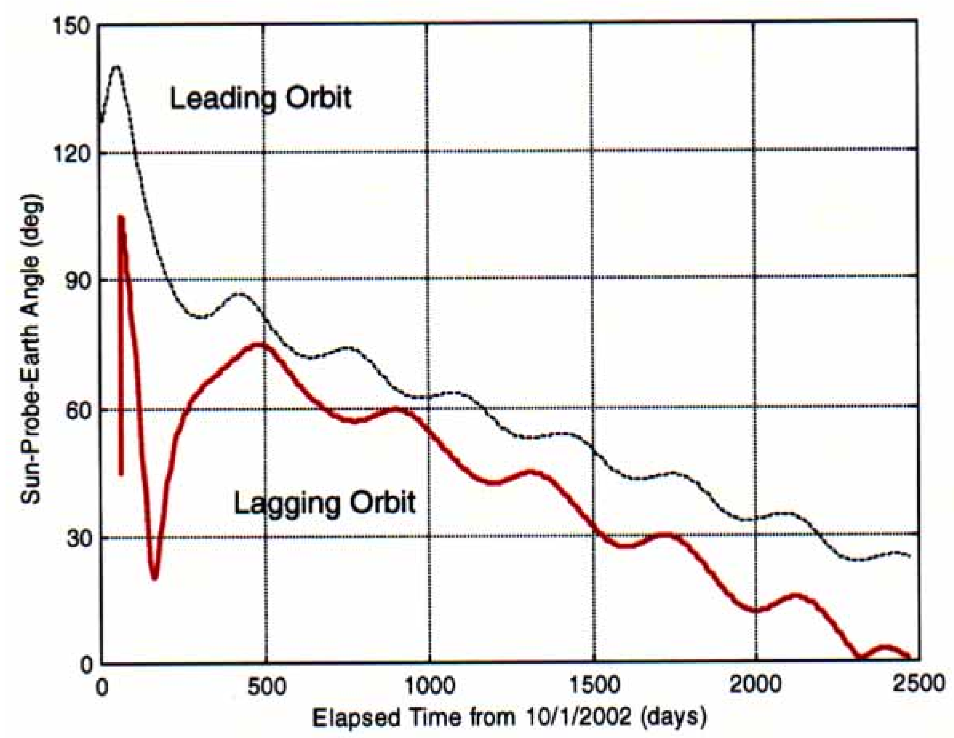


Figure 2-20 Sun-Probe-Earth Angle

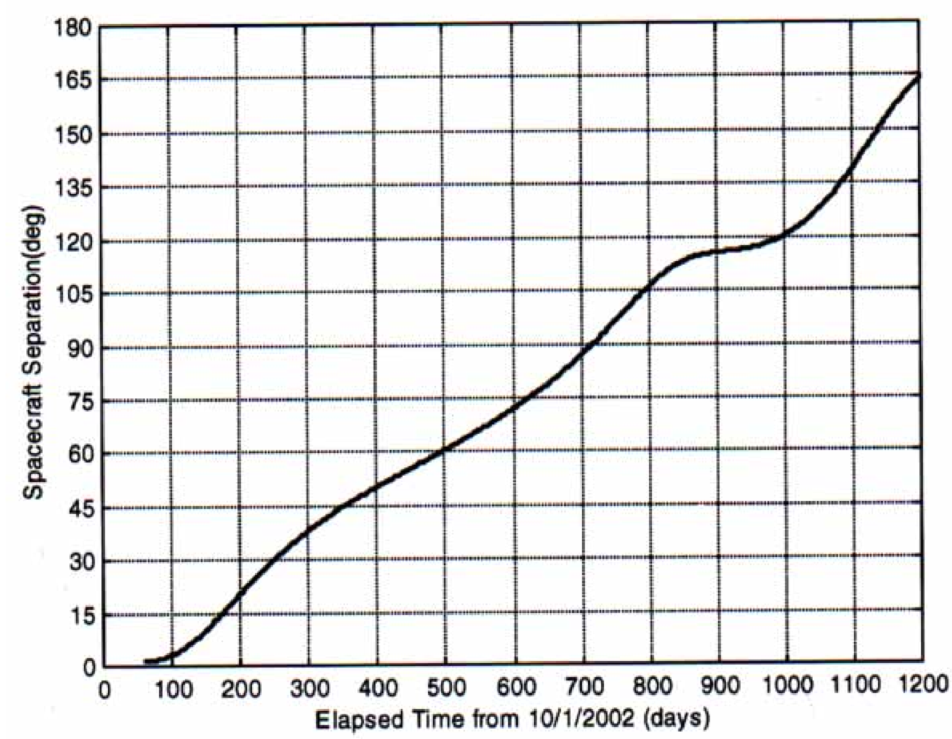


Figure 2-21 Angular Separation